DEVELOPMENT OF A COMPOSITE SATELLITE STRUCTURE FOR FORTÉ

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ABSTRACT

The Los Alamos National Laboratory (LANL) in partnership with Composite Optics Incorporated (COI) has advanced the development of low-cost, lightweight, composite technology for use in small satellite structures, in this case, for the Fast On-Orbit Recording of Transient Events (FORTÉ) satellite mission. The use of advanced composites in space applications is well developed, but the application of an all-composite satellite structure has not been achieved until now. This paper investigates the application of composite technology in the design of an all-composite spacecraft structure for small satellites. Engineering analysis and test results obtained from the development of the spacecraft engineering model are also presented.

INTRODUCTION

A common practice for constructing small spacecraft structures is to use an all-aluminum, rather than composite, spacecraft structure. This practice reduces the payload capacity significantly; however the cost of the aluminum structure has historically been lower than one that uses advanced composites. LANL mission requirements dictated the need for a long-term solution that substantially increased the ratio of payload to structural mass while maintaining a low-risk, low-cost approach. LANL intends to use the concept developed for FORTÉ on future missions requiring similar enhanced payload capacities.

The satellite program FORTÉ is the second in a series of satellites to be launched into orbit for the US Department of Energy. The FORTÉ program objective is to record atmospheric bursts of electromagnetic radiation. This paper will discuss the issues of design, analysis, and testing required to deliver the spacecraft and its associated components within a two-year period. The spacecraft will be launched into low earth orbit in 1996 from a L1011/Pegasus-XL launch vehicle. Due to the tight time constraints, a novel, low-cost solution using graphite fiber reinforced plastic composites was required to achieve the performance goals of the mission. The details of material selection, characterization of design allowables, analysis, test program and the approach used in determining the structural geometry that provide the optimum performance for this mission are presented. The resulting design for the structure is shown in Fig 1.

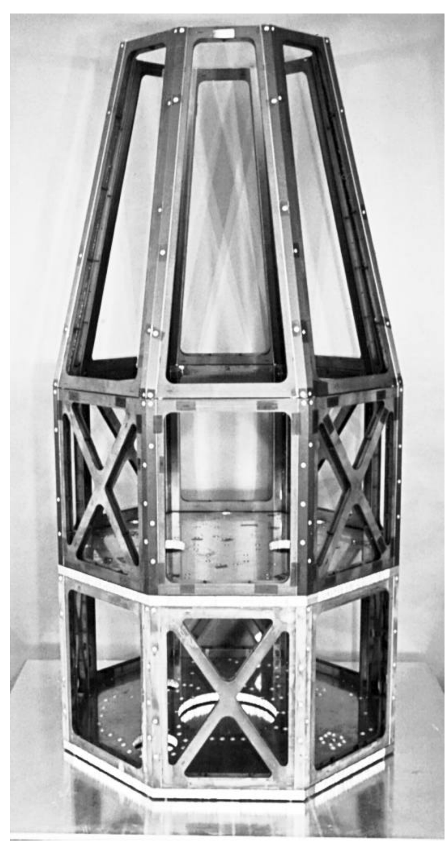

Figure 1. Spacecraft Structure

STRUCTURE DESIGN

Current practice in small satellite design uses conventional materials to reduce risk and cost of the mission. Because of the inherent high risk, large potential payoff, and short time line, it was appropriate for Los Alamos to initiate the development of an all-composite structure. FORTÉ will provide the test bed and space validation for this type of structure and for other key aspects of our technology that will be used in other space programs. This major technology development will make a significant contribution to the many industrial pursuits that involve advanced structural technology.

Several factors influenced the FORTÉ design. The approach used by LANL was to do a sufficient amount of analysis to validate the design concept and to thoroughly benchmark the concept through rigorous testing of the spacecraft. The schedule permitted two design iterations that allowed the Engineering Model (EM) to be thoroughly tested and subsequent changes to be fed back into the Flight Article (FA) that was constructed in the spring of 1995. The geometry is simple and modular for low cost, improved maintainability, and repairability. Finally, materials that are critical to the project's success have already been proven in space.

DESIGN CONSIDERATIONS

The basic spacecraft configuration was dictated by the Pegasus-XL. The octagonal shape of the spacecraft lends itself to using a modular construction. Because the cost of developing and employing molded composite fabrication techniques is high, the SNAPSAT™* approach developed at COI was adopted.

The spacecraft has several design considerations that were addressed in addition to the standard structural issues. Many payload elements have relatively tight temperature constraints because of the requirements of electronic components and all payload components have to be electrically grounded. High separation shock loads require mitigating shock between the bus and launch vehicle. The cage-to-deck interface requires positive metal-to-metal contact, which permits a well-controlled interface and efficient load transfer through the structure. In addition the spacecraft was required to be RF shielded.

The resulting design drivers for the spacecraft structure are weight, strength, stiffness, and launch vehicle volume. The overall cost, schedule, and associated risks also have a significant influence on the design.

A comparison was conducted between FORTÉ, aluminum alloy, and conventional composite structures. From Table 1, we can see that FORTÉ technology gave us a lightweight and durable structure that is faster to produce and had a cost equivalent to similar structures manufactured by other methods.

Table 1. Manufacturing Comparison Matrix

Parameter	FORTÉ Structure	Aluminum Alloy Structure	Conventional Composite Structure
Material	graphite/epoxy	aluminum	graphite/epoxy
Material Cost	high	low	high
Material Advantage	stiffness/weight/strength	cost	stiffness/weight/strength
Material Form	flat stock panels	metal billet	molded parts
Manufacturing Process	water jet	machining	molding
Process Cost	low	moderate	high
Process Advantage	quick turn around time	established technology	customized geometry
Fabrication Time	10 weeks	16 weeks	30 weeks
Tooling Cost	low	low	high
Structure Weight	42.6 kg (94 lb)	64.4 kg (142 lb)	42.6 kg (94 lb)
Unit Fabrication Cost	$160 k	$133 k	$400 k

* SNAPSAT™ is a patent pending trademark of COI

SPACECRAFT STRUCTURE DESIGN DESCRIPTION

The FORTÉ spacecraft primary structure consists of 6 major structural components: 3 structural trusses and 3 structural instrument decks, along with 24 solar array substrate (SAS) panels as shown in Fig. 2. The fundamental principles behind this unique spacecraft design are simplicity, modularity, and interchangeability.

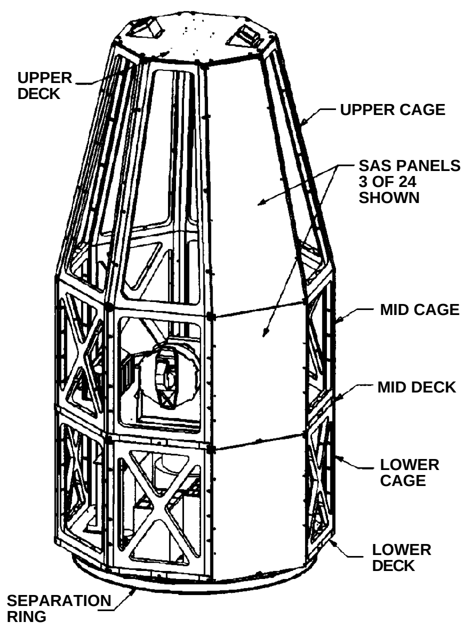

Figure 2. Structural Components

The three structural frame trusses are termed the lower, mid, and upper cages. They are all fabricated from flat stock graphite/epoxy (Gr/E). The lower and mid cages are identical to each other. Rectangular frame subassemblies comprise these two cages. The upper cage assembly is constructed using trapezoidal frame subassemblies. Eight frame subassemblies are bonded together to form each of the three octagonal cages.

The three decks are termed the lower, mid, and upper decks. The lower and mid decks are structurally similar to each other and will be used to mount most of the payload and bus components required for the FORTÉ mission. The upper deck closes out the structure. All three decks are fabricated from aluminum honeycomb sandwich-bonded between Gr/E skins.

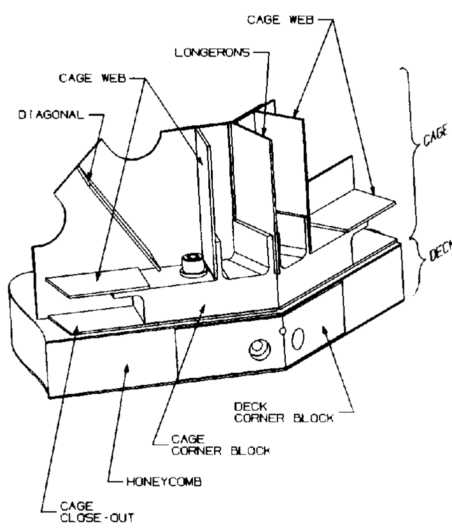

Figure 3. Structure Corner Fittings

The decks and cages are mechanically fastened to each other via aluminum corner fittings that are bonded into the cages and decks as shown in Fig. 3. This arrangement ensures that the structure has adequate load transfer in the highly loaded corners of the cage.

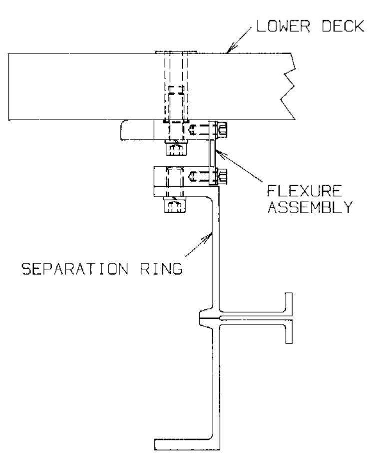

Figure 4. Cross Section Through Separation Ring and Flexure

Due to expected high shock loads when the satellite is separated from the booster, a shock mitigation system was implemented to protect delicate instruments mounted on the lower deck. The shock mitigation system consists of twenty flexures (Fig. 4) that connect the lower deck to the Pegasus-XL separation ring. These flexures efficiently attenuate the high frequency energy transfer and accommodate the coefficient of thermal expansion differences between the Gr/E structure and the aluminum separation ring.

DESIGN ALLOWABLES

Material validation efforts were initiated to define design allowables in critical areas of the structure. The primary concerns were the high shear stress areas of the SAS panels, the shear stress between the Gr/E and the aluminum angular interface block corner joints, and the deck component insert pullout allowables.

SAS PANEL COUPONS

Because the SAS panels were one of the most critical areas of the structure, edge coupons were fabricated to define the design allowables loads. The coupons were designed to test the shear strength in-plane of the SAS panel corner since analysis showed the maximum shear force was in this direction.

Table 2. SAS Panel Corner Shear-Out Results

	Mean Ultimate Shear-Out kN (lbs)	Design Allowable Shear-Out kN (lbs)
All Coupons Combined	4.18 (940)	3.34 (750)
Non Thermal Cycled Coupons	4.46 (1003)	3.92 (881)
Thermal Cycled Coupons	3.90 (877)	2.96 (666)

The effects of thermal cycling on the bonded joints was also of interest. The spacecraft is to be maintained near room temperature during the launch phase, but it will be cycled from -65°C to 80°C five times prior to launch as part of its qualification testing. Therefore it was imperative to know the effects of thermal cycling on the shear-out design allowable. Ten coupons were tested after thermal cycling and ten coupons were tested without thermal cycling.

The results of the two tests are summarized in Table 2. The average ultimate shear-out load for the thermal cycled coupons degraded by 13% and the design allowable was decreased by 24%.

SHEAR COUPONS

Another critical area for which little design data existed was the cage structure corners where aluminum angular interface blocks were bonded to the Gr/E skins. Initially the published shear strength for the adhesive was used to determine the design allowable. Fifteen single lap shear coupons were fabricated and tested. Of the fifteen, five were not thermally cycled and ten were subjected to the same thermal cycle as the corner coupons. The results of the shear coupon testing are shown in Table 3. The mean ultimate shear load showed no dependence on thermal cycling. The low value for the non-thermal-cycled set is a reflection of the small sample set size, given that the mean and standard deviation are almost identical to other cases.

Table 3. Cage Structure Corner Shear Coupon Results

	Measured Bulk Area Mean Ultimate Shear Stress MPa (psi)	Calculated Bulk Area Allowable Shear Stress MPa (psi)
All Coupons Combined (15 Coupons)	6.17 (895)	3.50 (507)
Non-Thermal-Cycled (5 Coupons)	6.12 (888)	1.59 (231)
Thermal - Cycled (10 Coupons)	6.21 (900)	3.02 (438)

Studies by Ojalvo and Eidinoff (1977) show that for a single lap shear joint the actual stress distribution varies considerably from the center to the edge of the joint. This distribution is also a strong function of the adherend thickness. A detailed analysis of the cage joint revealed that since the Gr/E skins were bonded to thick aluminum blocks and that all joints were closed with a face sheet, no rotation existed as with the single lap shear joints. The stress was evenly distributed. The bulk area shear was determined to be 2.23 MPa (324 psi) with a peak peel stress of 3.56 MPa (517 psi). These results are much lower than the design allowables shown in Table 3. Compared to the analytical results of the FEAs, the peak stresses were considered acceptable.

DECK INSERTS

The last area where testing was required to provide accurate design data was for the deck inserts. A deck coupon with twelve inserts was made and tested for pull-out and shear-out. The inserts in the decks do not use any corefil to secure them. Without the use of corefil, it was not necessary to thermal cycle these coupons. The results of this test are shown in Table 4.

Table 4. Deck Inserts Pull-Out Results

	Mean Ultimate Load kN (lbs)	Allowable Load kN (lbs)	Maximum Calculated Load at Launch kN (lbs)
Pull-out	2.93 (659)	2.19 (493)	0.64 (143)
Shear-out	3.90 (877)	2.94 (660)	0.70 (158)

STRUCTURAL STATIC ANALYSIS AND TESTING

The static analysis and component testing effort focused on optimizing the structure and developing design allowables for critical structural components. The main parameters that had to be optimized were the placement of cage cross bracing, the number of attachment points for the SAS panels, the design strengths of the SAS panel bolted joints, and the lap shear strength of the cage corner. The SAS panels were designed to be load carrying panels augmented with cage cross bracing. The fully assembled EM structure was then statically tested to corroborate the design and analysis, and to check for quality of workmanship.

The structure was modeled using the COSMOS FEA package. Three dimensional beam elements were used to simulate the longerons formed when the cage and decks are assembled. The decks were modeled using isotropic plate elements. The combined mechanical properties of the aluminum honeycomb with Gr/E skins were calculated and used as input for the decks and the SAS panels. The mass of the components was uniformly distributed over the entire surface area of the decks. The bolted attachment of the SAS panels was simulated using short three dimensional beam elements which transferred the bending of the longerons to the SAS panels. The model was rigidly restrained at the lower deck.

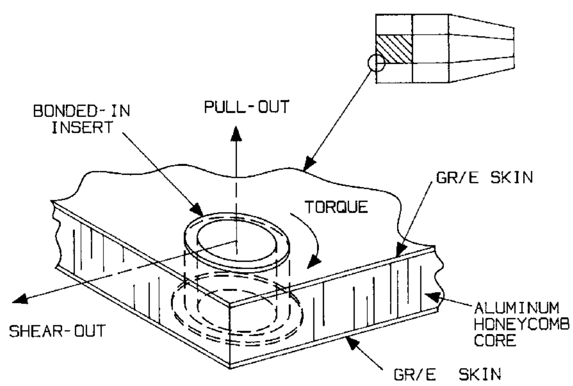

Figure 5. SAS Panel Maximum Loads

One of the most severe environments for the structure will be during the short time duration when the Pegasus-XL launch vehicle is dropped from the L1011. The structure was designed for accelerations of 8.5 g at the bottom deck and 18.5 g at the top deck. Since component access is critical, the number of cage frames with cross bracing had to be minimized and the SAS panels were required to carry the majority of the loads. The analysis showed that the high stress area for the structure was the corner indicated in Fig. 5. To minimize the forces at this corner the four openings in the bottom and middle cages must have cross braces and ten fasteners were required to meet the shear load for the SAS panels. The optimum location of the bracing was determined to be perpendicular and parallel to the drop plane for the lower cage and rotated forty-five degrees for the middle cage (Fig. 2). This would ensure the in-plane shear in the SAS panels would be below the design allowable shown in Table 5. The maximum calculated shear-out is 1.18 kN (265 lb), well below the design allowable of 2.96 kN (666 lb).

Table 5. SAS Panel Resultant Loads

Load	Maximum Calculated	Design Allowable
Pull-Out	0.02 kN (5 lb)	2.30 kN (516 lb)
Shear-Out	1.18 kN (265 lb)	2.96 kN (666 lb)
Torque-Out	0.56 N-m (5 in-lb)	8.20 N-m (72.6 in-lb)

Forces from the overall structural model were then used in a detailed model of the cage corner joint (Fig. 6). The joint is built up by bonding quasi-isotropic Gr/E skins to an aluminum block that is bolted to the decks. The model was constructed to calculate adhesive stresses and prying stresses at the bonded interfaces. The maximum in-plane stress in the adhesive is 2.23 MPa (324 psi) and the maximum longeron-aluminum block peel or prying stress was found to be 3.56 MPa (517 psi), below the allowables shown in Table 3.

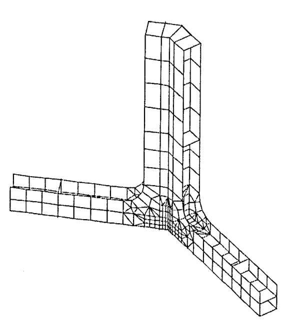

Figure 6. Finite Element Model of the Joint

Table 6. Joint Results

Longeron-Aluminum Block Max In-Plane Shear Stresses	2.23 MPa (324 psi)
Longeron-Aluminum Block Maximum Peel Stresses	3.56 MPa (517 psi)
Maximum Outer Skin Von Mises Stress	17.24 MPa (2500 psi)

Detailed testing of the SAS panels was performed to verify that the proper mechanical properties had been used in the overall model. The first five natural frequencies were calculated. These frequencies and modes were compared to values found from modal tests of the actual panels. The panels were subjected to a sine sweep and to determine the first five natural frequencies. The panels were then continuously subjected to this frequency and sand was used to identify the nodal points of the mode shape. A comparison of the calculated frequencies and the measured frequencies is shown in Table 7. The mode shape for the first natural frequency is shown in Fig. 7.

Table 7. Analytical and Measured Result for Fundamental Mode Shapes of a Type A SAS Panel

Mode #	FEA	FRF	% Difference
1	164.8	165	0.1%
2	203.3	214	5.0%
3	349.1	373	6.4%
4	375.5	389	3.5%
5	456.2	483	5.5%

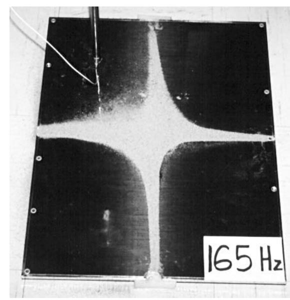

Figure 7. Measured Fundamental Frequency of a Type A SAS Panel

The fully assembled structure was statically tested to verify the design and check workmanship. The static design loads were applied to the structure at four places as depicted in Fig. 8. Displacements were measured at the same locations in both the load direction and perpendicular to the direction of the test load. This procedure was repeated four times, rotating the structure ninety degrees between each test. Displacements at the four locations versus percent full scale test load for a typical test are shown in Fig. 9. Displacements increase nonlinearly as the load increases due to slippage of the bolted joints between the SAS panels, the cages and decks, and because of warping of the shock mitigation flexures. As the test load is relaxed the hysteresis is caused once again by the resistance to slippage at the bolted joints. The maximum deflection at the top deck was measured to be 8.0 mm (.315 in) while static test predictions from the model were 6.99 mm (.275 in).

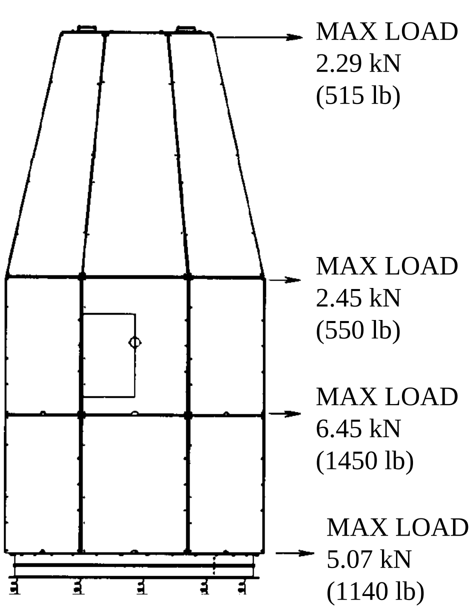

Figure 8. FORTÉ Spacecraft Loading During Drop

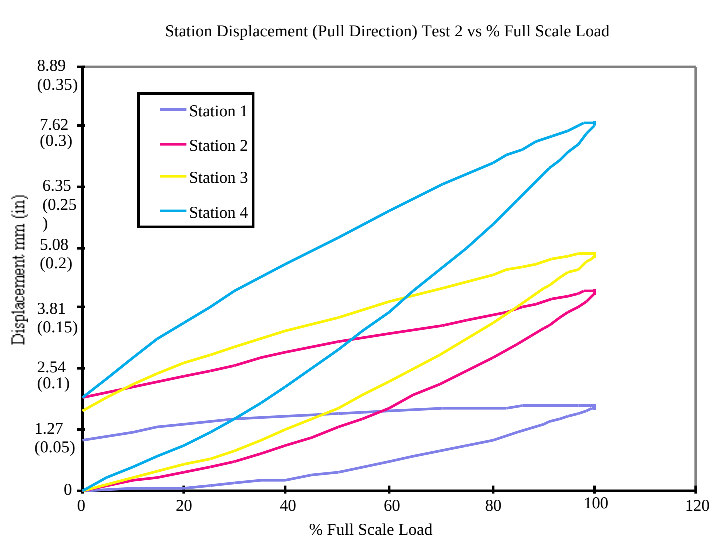

Figure 9. Displacement vs. Percent Full Scale Test Load

STRUCTURAL DYNAMIC ANALYSIS AND TESTING

In designing the FORTÉ satellite structure three significant dynamic launch events were considered. The first was the random vibration environment that loads the satellite from the time the L1011 aircraft takes off until the third, and final, stage of the Pegasus launch vehicle completes its burn. A second transient event occurs when Pegasus-XL is "dropped" from the carrier aircraft. This short duration event loads the satellite in the lateral direction as the launch vehicle straightens from the slightly bowed condition that it experiences during captive carry. Finally, the third transient event is the high-frequency shock load that affects the satellite when it is separated from the third stage of Pegasus-XL.

STRUCTURAL VIBRATION MODES

The dynamic response characteristics of the satellite are affected by its relatively high stiffness and its low damping. These characteristics were determined through analysis with the ABAQUS FEA computer program and verified with a modal test of the structure with mass simulators attached representing satellite bus and payload components. The most important vibration modes are summarized in Table 8. The fundamental deck modes all occur at frequencies below the first overall axial bending mode of the satellite. This fact leads to high amplification for these modes and, therefore, potentially excessive loads on the attached components.

Table 8. Vibration Modes of FORTÉ Satellite

Mode Shape Description	Frequency (Hz)	Damping (%)
1st bending (y-direction)	42.4	1.3
1st bending (x-direction)	42.8	1.3
1st bending (lower deck)	45.0	1.2
1st bending (mid deck)	67.5	0.9
2nd bending (lower deck)	71.0	
2nd bending (lower deck)	75.8	
1st bending (top deck)	90.2	
1st torsion	124.3	0.6
2nd bending (y-direction)	124.9	2.4
2nd bending (x-direction)	125.6	2.1
2nd bending (mid deck)	156.5	
2nd bending (mid deck)	160.7	
1st overall axial mode	173.4	1.0

Correlation of the FEA with the measured mode shapes and frequencies required reducing the stiffness of the SAS panels to account for their local compliance at the points where they attach to the main structure. To accomplish this correlation the stiffness was reduced between a factor of four and eight at the attachment points. The orthotropic elastic properties of the honeycomb for the two main decks were also adjusted to obtain the measured modal frequencies for the first deck bending modes.

RANDOM VIBRATION

Fig. 10 shows the FORTÉ EM attached to one of the Los Alamos shake tables in preparation for a random vibration test. Table 9 gives the power spectral density (PSD) that envelopes the expected vibration environment during the complete launch sequence. The EM of the FORTÉ structure with mass simulators attached was tested to this environment in three directions. The first vibration test sequence identified a problem involving high response near the centers of both the mid and upper decks. Response PSDs were particularly high at frequencies representing the fundamental bending modes of the decks. Two approaches were used to mitigate these high responses. On the upper deck some of the components were moved to brackets on the cage structure next to the deck. Other components were mounted on low-frequency, vibration isolation brackets. Components could not be conveniently repositioned or isolated on the mid deck so a different approach is being considered there. At the time of this writing several different methods utilizing viscoelastic materials are being considered for adding damping to lower the mid deck response.

Figure 10. Random Vibration Test Facility

DROP TRANSIENT

The high stiffness of the FORTÉ structure was beneficial in limiting loads on the components during the drop transient. That transient has little energy above 20 Hz and with the first bending modes of the satellite being above 40 Hz, little amplification was expected. The EM was tested at levels twice the value of the expected levels during launch. During this test the response levels on the satellite were approximately 6 to 8 g for components on the lower deck, 8.5 g at the mid deck, and 13 g at the upper deck.

Table 9. Protoqual Random Vibration Spectrum

20 - 800 Hz	0.008 G^2/Hz
800 - 1000 Hz	+7.55 dB/Octave
1000 - 1300 Hz	0.014 G^2/Hz
1300 - 2000 Hz	-13.61 dB/Octave
2000 Hz	0.002 G^2/Hz
Overall Level	4.08 G_{rms}

CONCLUSIONS

LANL and COI have designed, fabricated, analyzed, and tested a simplified, cost-effective method for the production of small satellite spacecraft structures. This process produces an all-composite spacecraft structure that is lightweight and strong, providing substantial improvement over aluminum designs in its payload-to-weight ratio. The fabrication technology that has been developed produces savings in production time over conventional composite processes. The simple but robust spacecraft structure provides a platform that will be useful for a wide variety of applications.

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REFERENCES

Goland, M. and E. Reissner, “The Stresses in Cemented Joints”, Journal of Applied Mechanics, March 1994, Vol. 11, pp. A17-A27.

Griffin, M. D. and J. R. French, *Space Vehicle Design*, AIAA Publishing, 1991, pp. 345, 351, 356.

“Mechanical Properties of Hexcel Honeycomb Materials”, Hexcel TSB 120, Hexcel Corporation, 1987.

Ojalvo, I.U., and H. L. Eidinoff, "Bond Thickness Effects Upon Stresses in Single-Lap Adhesive Joints”, AIAA Journal, Vol.16, No.3, 1977.

Pegasus Air Force Small Launch Vehicle Interface Design Document, Release 1.00, August 31, 1993.

Thompson, T. C., et. al., "Development of an All-Composite Spacecraft Structure for Small Satellite Programs", Los Alamos National Laboratory, LA-UR 94-2897, 1994.